PAPER NUMBER 59-

Propulsion Requirements for Space-Station Erection

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The rapid advances in rocket and space technology will lead eventually to the establishment of a manned space station. The first true, manned space station probably will be assembled in an orbit around the earth. The personnel for construction of the space station, as well as the materials of construction, will have to be transported to the desired location in space by a rocket vehicle, probably launched from the earth. During the construction phase of the space-station erection, there will be a need for auxiliary propulsion power units for transferring both personnel and material from the launching vehicle to the construction site. For the purposes of this paper, only propulsion systems which are applicable to the actual construction phase of the space station will be considered. The propulsion requirements for reaching the desired position in space where the station is to be erected is beyond the scope of this paper. Some of the more important requirements which must be met by a propulsion system which is to be utilized in the erection of a space station are: (a) Safety and reliability; (b) versatility; (c) high performance.

THE AMERICAN SOCIETY OF MECHANICAL ENGINEERS
29 West 39th Street, New York 18, N. Y.

For presentation at the Aviation Conference, Los Angeles, Calif., March 9-12, 1959, of The American Society of Mechanical Engineers. Manuscript received at ASME Headquarters, January 26, 1959.

Written discussion on this paper will be accepted up to April 15, 1959.

Copies will be available until January 1, 1960.

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Since the propulsion systems for a manned space station must operate in the vicinity of personnel, safety and reliability are of prime importance. Every precaution would have to be taken to prevent any malfunction of the system which might result in a rupture of a propellant tank, line, or valve or an explosion in the thrust chamber, any of which could result in injury to nearby personnel.

Versatility would be important because of the variety of tasks that would have to be performed by a limited number of units. The applications of the propulsion system in the erection of a space station would range from moving personnel from point to point, to the transfer of materials of construction and relatively large pieces of equipment. The required thrust for transporting the different items at the required rate would necessitate a substantial range in thrust level. For that reason it would be desirable that the propulsion systems have variable thrust levels so that a larger variety of tasks could be performed with fewer units. That feature would result in a substantial savings in weight in launching the systems into orbit.

High performance would be desired to reduce the weights of both the hardware and the propellants that would have to be transported from the earth to the desired location in space.

In a consideration of the requirements of safety and reliability, versatility and high performance, the storable, hypergolic, liquid-propellant systems offer definite advantages over other types of propellants. The advantages of a propellant system which is storable are evident.

The common concept of "storability" is altered slightly when considering propulsion systems for outer space. Cryogenic materials which are normally considered "non-storable" in the environment on earth may possibly be found to be "storable" under conditions existing in outer space. This broadens the base for selection of propellant materials. Hypergolicity is a necessary characteristic to promote safety and reliability in the thrust-chamber operation. In addition, it simplifies the propulsion-system design by eliminating the need for ignition systems; hypergolic ignition avoids the formation of potentially dangerous mixture of fuel and oxidizers in cases of igniter failure. This requirement of hypergolicity rules out propellants such as liquid oxygen and liquid hydrogen and leaves nitrogen-oxide-base oxidizers and hydrazine-base fuels as the current major contenders for the propulsion-system propellants.

The importance of safety and reliability in the operation of the propulsion system combined with the requirement of high performance suggests a new concept in propulsion-system design. That concept involves operating at ultra-low combustion pressures (5-10 psia). The following discussion will illustrate the advantages of low chamber-pressure operation as well as the problems that might be encountered.

LOW CHAMBER-PRESSURE OPERATION

As indicated previously, because of the necessity of the proximity of the rocket power plant and personnel during the construction phase of a space station, safety of operation of the propulsion system is of prime importance because of the destructive nature of rupturing tanks, lines, valves, and so on. Operating at low chamber pressures would tend to reduce the hazards connected with ruptures occurring in the system components. For instance, in operating a chamber pressure at 5 psia, the pressures in the propellant tanks, lines, and valves would probably not exceed 15-25 psia (in the range of a conventional gasoline stove) compared to pressures of 400-500 psia in those components when operating at chamber pressures of the order of 300 psia. The danger to nearby personnel in the advent of the rupture of a component operating at a 15-25 psia internal pressure would be appreciably less than operating at pressures in the 400 to 500-psia range.

Operating at a low chamber pressure would permit the use of simplified techniques to cool the thrust chamber as well as the application of a simplified type of propellant-feed system, both of which contribute to reliability through a simplification in the system design. The design and operation of a thrust chamber at a combustion pressure in the 5 to 10-psia range may introduce problems resulting from the effect of low pressures on combustion. Design criteria which are suitable and accepted for present-day rocket thrust chambers may have to be revised for low-pressure operation. Some of the possible problem areas are considered in the following discussion.

Combustion

Ignition. One of the most critical stages of the operation of a rocket power plant is ignition. For hypergolic propellants, delayed ignition allows the accumulation of propellants in the combustion chamber until ignition does occur. The ignition of the accumulated propellants results in a large pressure surge which, in the extreme, can destroy the thrust chamber. There is experimental evidence that pressure has an effect of ignition limits of certain propellant combinations. Some of the hypergolic propellant systems which have short ignition delays at atmospheric pressure become difficult to ignite at altitude conditions. Experience has shown that for operating at sea-level conditions, ignition delays of hypergolic propellants should generally not exceed 25 millisec. It is not known whether the same is true for operation in a vacuum.

Flame Speeds. Another aspect of low chamber-pressure operation is the effect of combustion pressure on flame speeds. If low pressures caused a reduction in flame speeds, larger combustion chambers might be required to attain complete combustion. Experiments on low-pressure combustion have been confined mainly to fuel-air mixtures; virtually no experimental data are available on the higher-energy systems discussed herein.

Some general effects of combustion pressure on flame speeds may be surmised. Albright, Heath, and Thena have shown that both increases and decreases of flame speed with pressure may be found for the same mixture according to the methods by which the flame speed is measured (1).[1] Apart from its influence on the number of molecular collisions per unit volume, pressure influences the equilibrium temperature of the burned gases by altering the degree of chemical dissociation, thereby altering the enthalpy change of the reaction. The pressure also influences the concentration of particles such as H-atoms. These temperature and concentration changes may be expected to cause changes in flame speed.

The effect of turbulence on flame speed is quite pronounced. Laminar flame speeds are often increased ten-fold by the introduction of turbulence. Since the turbulence level is affected by the Reynolds number of the gases in a combustion chamber, it is expected that the operating pressure, because it is involved in the Reynolds number, would have an effect on the flame speed. Reducing the combustion pressure would reduce the Reynolds number (and hence the turbulence level), assuming no effect on other parameters.

Performance. There will be a premium on performance and safety of operation in the selection of propellants for space-station propulsion systems. High performance will be at a premium since the propellants will have to be transported into the orbit in which the space station is to be erected.

Hypergolic storable propellants are currently available that exceed 300 lb/sec/lb specific impulse for 25 to 1 nozzle expansion ratios to a vacuum from a 500 psia chamber pressure. A group of propellants including N_2O_4/UDMH and N_2O_4/N_2H_4 give essentially the same performance under 25/1 expansion ratios to a vacuum. A choice from this group can then be based on factors such as ease of handling, reliability, chemical storability, and insensitivity to space radiations.

Low chamber-pressure operation will result in a decrease in the combustion performance, according to theoretical calculations. However, this reduction in thrust-chamber performance is partially counterbalanced by the reduction in the propulsion-system component weights and further compensated for by the increase in the propulsion system safety.

Thrust-Chamber Design - Heat Transfer

One of the major considerations in the design of a rocket thrust chamber is that of maintaining the walls at a safe operating temperature. Four general methods are employed for thrust-chamber cooling:

1 Regenerative cooling.
2 Heat-sink cooling.
3 Transpiration cooling.
4 Radiation cooling.

Probably the most widely used technique for chamber-wall cooling is regenerative cooling. (1)
The regenerative-cooled thrust chamber employs one or both of the propellants flowing through coolant passages in the chamber walls to remove heat transferred to the walls from the hot combustion gases. The maximum amount of heat which can be absorbed by regenerative cooling is limited by the velocity and the vapor pressure of the coolant. If the coolant velocity is too low, the rate of heat removal from the walls is insufficient to maintain the walls below their

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[1] Underlined numbers in parentheses designate References at end of paper.

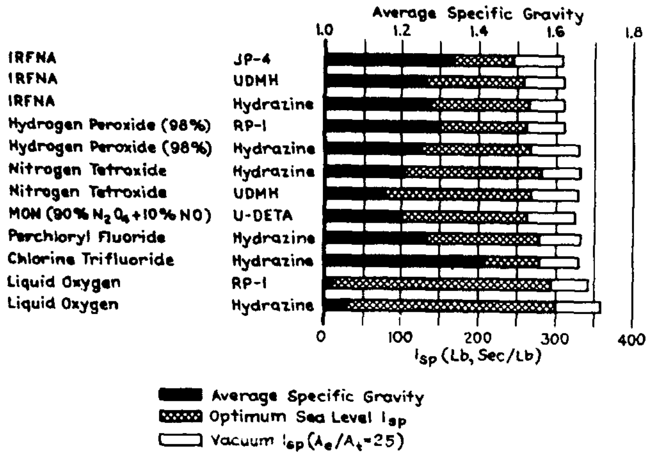

Fig.1 Comparative performance of various propellant systems P_c = 700 psia, optimum mixture ratio, shifting equilibrium.

rupture point. On the other hand, if the pressure in the coolant passages is not maintained at a high enough value, the fluid will vaporize and the burn-out heat flux will be reduced. Hence, a prime requirement for successful regenerative cooling is sufficient pressure drop in the coolant passages to maintain the desired coolant velocities plus a high enough pressure level to prevent the coolant from vaporizing. That requirement results in higher pressures in the propellant tanks, valves, and so on, which in turn requires an increase in the weight of those components. In addition, the coolant passages around the thrust chamber increase the weight of the system. Another problem in regenerative cooling is encountered in the use of fuels which are unstable at elevated temperatures. For instance, hydrocarbon fuels employed as a regenerative coolant, tend to break down chemically and form a layer of carbonaceous material on the walls of the coolant tubes. The layer of carbonaceous material acts as a thermal insulator and causes an appreciable reduction in the rate of heat transfer to the coolant. As another example, hydrazine when heated to a high temperature will decompose explosively. Hence, in the use of that fuel as a regenerative coolant, proper precautions must be taken to prevent its temperature from becoming too high.

The heat-sink method of cooling is based on the chamber wall material absorbing the transferred heat. This method of cooling is generally limited to short-duration, thrust-chamber operation. Current thrust chambers are fabricated of metals such as stainless steels which have relatively low specific heat. Hence, a relatively high mass must be used to absorb the heat rejected by the hot combustion gases as during a run, the wall temperature rises continually. If the run duration is too long, the chamber wall will begin to melt and ultimately a burn out occurs. For the application proposed herein the heat-sink technique would be unsatisfactory.

The third cooling method, usually referred to as film cooling, consists of injecting a fluid into the combustion chamber at one or more locations in such a manner that it flows along the walls forming an insulating film of liquid or vapor. Transpiration, or sweat-cooling, are names applied to the techniques where the chamber wall is porous so that the coolant can be injected over the entire surface of the wall to form a continuous protective layer.

The present state of the art indicates that, though transpiration cooling may be an effective method of cooling rocket thrust chambers, the lack of experience with this technique precludes its immediate application.

The fourth method of cooling is radiation cooling. Radiation cooling depends on the ability to remove heat by radiation from the outside of the thrust chamber walls at a rate that is sufficient to keep the walls below their maximum operation temperature. That rate of heat transfer is determined by the allowable wall temperature and the emissivity of the material forming the thrust chamber. The feasibility of using radiation heat-transfer-cooling a thrust chamber operating at a low combustion pressure has been given an extensive treatment by Lt. Drea Hast of the Propulsion Laboratory, Wright Air Development Center, as well as by other investigators.

Heat Transfer from Hot Gases. The coefficient of heat transfer for the gas side of a rocket thrust chamber may be effectively calculated in most cases by the equation

$$h_g = 0.0295\, C_p G^{0.8} Pr^{-2/3} (\mu_f/D)^{0.2} (\rho_f/\rho_b)^{0.8} \tag{1}$$

where

h_g = gas-side coefficient of heat transfer
C_p = constant-pressure specific heat of gas film
G = mass velocity of bulk gases
Pr = Prandtl number of gas film
μ_f = viscosity of gas film
D = chamber diameter
ρ_f = density of film gases
ρ_b = density of bulk gases

C_p, Pr, μ_f, and ρ_f are usually evaluated at

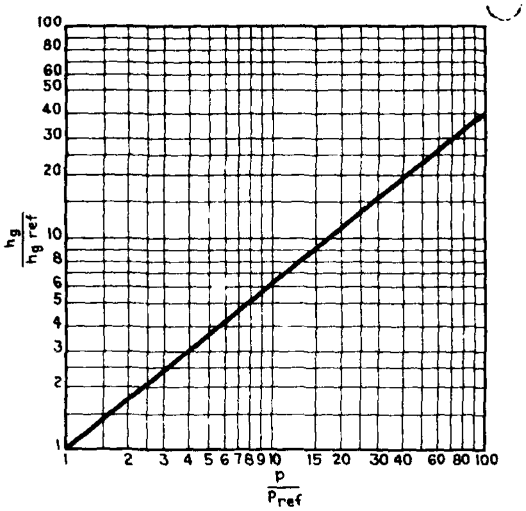

Fig.2 Effect of combustion pressure on coefficient of heat transfer.

the average film temperature. The NACA has found that, with air at a high Reynolds number and high temperatures, the Reynolds number evaluated by $(GD/\mu_f)(\rho_f/\rho_b)$ correlated considerably better than by GD/μ_f.

The term ρ_f/ρ_b may be evaluated by the equation (2)

$$\rho_f/\rho_b = 2T_c/(T_r + T_w)\left\{1/\left[1-(\gamma + 1)M^2/2\right]\right\}$$

where

T_c = combustion temperature

T_r = recovery temperature at chamber wall

T_w = wall temperature

M = Mach number of combustion gases

With a calculated value of h_g, the heat flux into the wall may be obtained by

$$q/A = h_g(T_r - T_w) \qquad (3)$$

For fixed values of T_r and T_w, the heat flux into the thrust-chamber wall is dependent only on the magnitude of the heat-transfer coefficient.

In equation (1), the main influence of pressure on the coefficient of heat transfer is the term $G^{0.8} = (\rho V)^{0.8}$.

Assuming the perfect gas law, ρ should vary directly with the combustion pressure. Then

$$q/A \propto h_g \propto G^{0.8} \propto \rho^{0.8} \qquad (4)$$

Proportionality (4) indicates that the heat flux through the gas film should vary with the 0.8 power of the chamber pressure. Fig. 1 is a plot

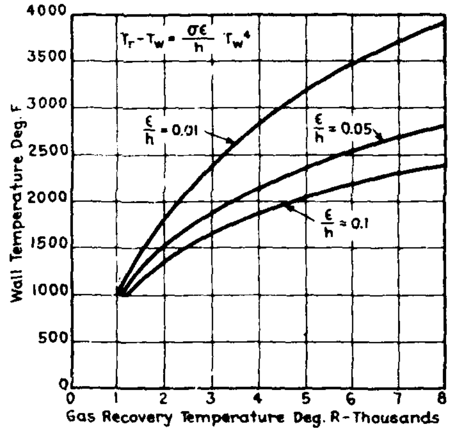

Fig.3 Thrust-chamber wall temperature versus gas recovery-temperature for radiation cooled thrust chamber.

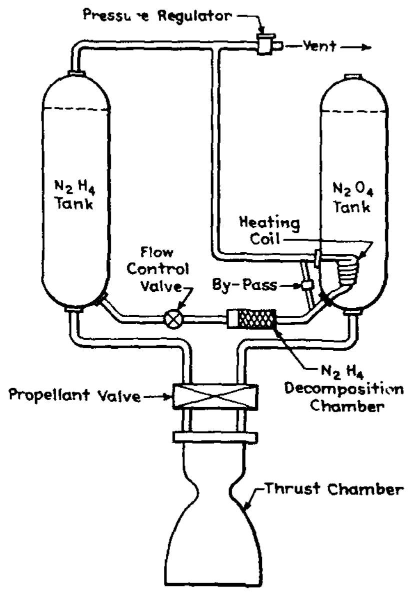

Fig.4 Vapor-pressure feed system.

of $h_g/h_{g\,ref}$ versus p/p_{ref} on logarithmic graph paper where $h_{g\,ref}$ is the value of the gas side heat-transfer coefficient at some reference pressure, p_{ref}.

In referring to that plot it may be seen that reducing the chamber pressure by a factor of 100, i.e., from 500 psia to 5 psia, will theoretically reduce the heat-transfer coefficient by a factor of approximately 40.

Values for the gas-side coefficient of heat transfer in a thrust chamber operating at a combustion pressure of 500 psia are of the order of 500 Btu/hr-ft^2 - deg F. Hence, if the combustion pressure is reduced from 500 to 5 psia, the coefficient of heat transfer for the gas film would theoretically be reduced from a value of 500 Btu/ft^2-hr-deg F to a value of 12.5 Btu/ft^2-hr-deg F. A specific example will now be considered to illustrate the feasibility of radiation cooling at a combustion pressure of 5 psia.

Employing the aforementioned equations together with the equation for radiant heat transfer, the equilibrium wall temperature which will be reached for a radiation-cooled thrust chamber will be calculated assuming the following:

1 The combustion gas temperature and recovery temperature are equal.

2 The environmental temperature is 0 deg R.

3 The temperature gradient across the combustion chamber wall may be neglected.

With the foregoing assumptions, the temperature of the combustion chamber wall can be determined by equating the heat transferred into the wall from the hot gases (Newton's law of cooling) to the heat transferred away from the outside of the wall by radiation cooling (Stefan Boltzman equation)

$$q/A = h_g(T_r - T_w) = \epsilon \sigma T_w^4$$

where

q/A = heat transferred per unit area

σ = Stephan-Boltzman constant

ϵ = emissivity of the outside chamber wall.

Fig. 2 is a plot of thrust-chamber wall temperature versus combustion gas temperature for different values of $\epsilon \sigma/h_g$. Assuming a value of 15 Btu/ft^2-hr-deg F for the heat-transfer coefficient, and a value of 0.75 for the emissivity of the outside surface of the combustion chamber, the equilibrium wall temperature for a combustion temperature of 5460 R is seen from Fig. 2 to be 2420 R (1960 F). From this a conclusion can be drawn that radiation cooling of low-pressure thrust chambers is feasible and therefore they should be considered for auxiliary propulsion systems for use in space.

Systems Aspects - Vapor Pressure Feed Systems

To force the fuel and oxidizer into the thrust chamber, most liquid-propellant rockets use an inert-gas pressurization system, chemical-pressurization system, or a turbopump feed system. The inert-gas-pressurization system employs helium, nitrogen, or some other gas which is fed at a regulated pressure to the fuel and oxidizer propellant tanks.

The turbopump feed system employs a gas generator which produces hot gases to drive a gas turbine which in turn, runs fuel and oxidizer pumps. A turbopump system is considerably more complex and hence less reliable than an inert-gas feed system. The choice of an inert gas feed system or the turbopump feed system depends on the run duration and the thrust level. The inert gas feed system is preferable for low-thrust and high-thrust short-duration systems. As the thrust level and duration are both increased, the weight of an inert-gas feed system increases rapidly compared to the weight of a turbopump system. At some point the turbopump system becomes the preferred feed system.

The use of a very low chamber pressure makes possible the consideration of an entirely different type of pressurization system. This is a vapor-pressure feed system which utilizes the vapor pressure of the propellants to pressurize the propellant tanks. The advantage of such a system is that it eliminates the weight of a pressure vessel for containing pressurizing gas; it also reduces the volume of the pressurizing gas required. The requirements for a vapor-pressure feed system includes a source of heat for maintaining the propellants or pressurizing liquid at the temperature required to obtain the desired vapor pressure. As an example the N_2O_4/N_2H_4 system is discussed.

The saturation temperature of N_2O_4 at 25 psia (92 F) could be maintained prior to use of the system by controlling the amount of solar radiation reaching the outside surface of the tanks. To supply the necessary heat during operation and to produce pressurizing gas for the N_2H_4 a small amount of hydrazine is decomposed, and the hot decomposition gases are passed through a heat exchanger located in the N_2O_4 tank. The heat input to the N_2O_4 is controlled so as to be sufficient to maintain the required vapor pressure in that tank. The cooled N_2H_4 decomposition products are then employed to pressurize the N_2H_4 tank directly. The rate of heat transferred to the N_2O_4 in the heat exchanger is controlled by the by-pass valve and the flow rate of N_2H_4 to the decomposition chamber. The pressure in the N_2H_4 tank is controlled by regulating the flow rate of decomposed gases into the tank by means of a regulator. A portion of the gases can be vented if required.

Conclusions

To best fulfill the requirements of maximum safety, reliability, versatility and performance, auxiliary propulsion systems for use in the erection of a space station will operate with storable, hypergolic propellants. Low chamber-pressure operation offers attractive design features including safety, simplicity and reduced system component weight. Satisfactory solutions to the apparent problems associated with the design of these propulsion systems can be obtained by proper research and development effort.

References

1 R. E. Albright, D. P. Heath, and R. H. Thena, Industrial and Engineering Chemistry, vol. 44, 1952, p. 2490.

2 F. H. Garner and V. E. Henry, "Behavior of Sprays Under High Altitude Conditions," Fuel, vol. 32, 1953; p. 151.

3 W. H. McAdams, "Heat Transmission," McGraw-Hill Book Company, Inc., New York, N. Y. 1942.